

HEDRON

CORNICEN SPACECRAFT

Orbital Debris Assessment Report (ODAR)

Initial Release

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ORBITAL DEBRIS ASSESSMENT REPORT

This document contains the orbital debris assessment report (ODAR) for the Hedron Space Inc. (“Hedron”) (formerly Analytical Space, Inc.) experimental spacecraft Cornicen.

Table 1 includes a summary of the results in the following sections.

Table 1: Cornicen Launch Information

ODAR Requirement for Cornicen	Compliant or N/A	Not Compliant	Incomplete
4.3.1-a	x		
4.3.1-b	x		
4.3.2	x		
4.4-1	x		
4.4-2	x		
4.4-3	x		
4.4-4	x		
4.5-1	x		
4.5-2	x		
4.6-1(a)	x		
4.6-2	x		
4.6-3	x		
4.6-4	x		
4.7-1	x		
4.8-1	x		

ODAR SECTION 1: PROGRAM MANAGEMENT AND MISSION OVERVIEW

Project Manager: Katherine Monson

Foreign Government / Space Agency Participation: None

Schedule of Upcoming Mission Milestones: Launch scheduled for October 2022.

Mission Overview:

The Cornicen experimental spacecraft is scheduled for launch in October 2022 aboard a SpaceX Falcon 9 rocket. The spacecraft is expected to be inserted into a sun-synchronous orbit (SSO) with a target perigee and apogee of 550 km, with a 25 km orbital variance (i.e. between 525-575 km), an SSO inclination corresponding to the insertion altitude ± 0.1 degrees (approximately 98 degrees), and a local time of the descending node (LTDN) of 10:30 +30/-15 min. The spacecraft will maneuver from this insertion orbit to an operational 518 km circular SSO, with an inclination of approximately 97.5 degrees, at LTDN 10:30 ± 15 min. At this operational orbit, the vehicle will formation fly with a customer spacecraft, a Maxar satellite, maintaining a distance between approximately 50-100 km.

The mission will demonstrate experimental hardware and software developed by Hedron. Following these demonstrations, which are expected to be for approximately three years, the spacecraft will use the expected remaining propellant on board to lower its orbital altitude to below 400 km and then passively deorbit in approximately six months.

ODAR Summary:

1. No orbital debris to be released as part of normal operations
2. No credible scenarios for spacecraft breakup
3. Collision probability is compliant with NASA ODAR requirements 4.5-1 and 4.5-2 as calculated by DAS v3.1.2
4. Orbit decay lifetime due to atmospheric drag is less than 25 years and therefore compliant with ODAR requirement 4.6
5. Reentry debris casualty risk is compliant with NASA ODAR 4.7-1 as calculated by DAS v3.1.2

Launch Vehicle and Launch Site: SpaceX Falcon 9; Florida, USA

Projected Launch Date: October 2022

Mission Duration:

1. Nominal operations: 3 years
2. Post-operations orbital lifetime: approximately 6 months from an orbit altitude of 400 km (3.5 years from the 518km orbit altitude)

Launch and Deployment Profile:

The launch and deployment profile consists of launch and insertion into orbit, followed by the approach maneuver from the insertion orbit to the operational orbit. Although some operations are done during the approach maneuver, this approach is heavily related to the insertion orbit. Thus, it is discussed in this section. The Cornicen launch, insertion and deployment information is provided in Table 1.1.

Table 1.1: Summary of Cornicen Launch and Insertion, Operational Orbit Information

Target Launch Date	October 2022	
Launch Vehicle	SpaceX Falcon 9	
Secondary Payload Aggregator	Spaceflight Industries	
Cornicen Launch Configuration	Smallsat measuring 39 cm x 39 cm x 60 cm	
	Target Launch and Insertion	Operational
Perigee Altitude	550 km \pm 25 km	518 km \pm 1 km
Apogee Altitude	550 km + 25 km	518 km \pm 1 km
Inclination	SSO \pm 0.1° (~98°)	97.47° \pm 0.004°
LTDN	10:30 +30/-15 minutes	10:30 \pm 15 minutes
RAAN	Nominal +7.5°/-3.75°	Nominal \pm 3.75°

Launch and Insertion:

Spacecraft, as scheduled, will launch on a SpaceX Falcon 9 rocket in October 2022. Spaceflight Industries will integrate Cornicen as a secondary payload. The launch and insertion orbit details are provided in Table 1.1. For the provided insertion orbit range of 525 km SSO to 575 km SSO, the maximum insertion orbit altitude of 575 km corresponds to a 97.59 degree inclination at SSO and a period of 95.65 minutes.

Approach:

Following insertion, Cornicen will complete an approach maneuver from the insertion orbit to the targeted operational circular SSO at 518 km altitude, 97.47 degrees inclination, and 10:30 LTDN \pm 15 minutes. This approach maneuver is optimized for minimum maneuver time and dependent on the change in orbital parameters between insertion and operational orbits, including right ascension of the ascending node

(RAAN) difference, altitude difference, and inclination difference. While optimizing for minimum maneuver duration, it is possible that the approach maneuver moves the vehicle out of the altitude and inclination range between the insertion and operational orbits. This can result in a maximum altitude greater than either the insertion or operational orbit altitudes, resulting in a potential worst-case deorbit time scenario for a failure at this maximum altitude. This scenario is the focus of this approach maneuver section.

The approach maneuver can require an altitude raise to produce a negative RAAN change. In terms of maximum altitude raise, the worst-case insertion orbit RAAN change is a maximum negative RAAN change (*i.e.*, the RAAN needs to be decreased to reach the operational SSO from insertion) from 11:00 LT DN to 10:15 LT DN for a -11.25 degrees RAAN change. The worst-case insertion altitude for this maneuver is 575 km, as this leads to maximum altitude change to reach the 518 km operational orbit. The worst-case insertion orbit inclination for this maneuver is 97.69 degrees, which is the 575 km SSO inclination of 97.59 degrees plus a positive 0.1-degree error, for the maximum inclination change to reach the lower operational orbit inclination of 97.47 degrees.

This worst-case maximum altitude approach maneuver exceeds the mission altitude constraint of 645 km circular when optimizing for minimum maneuver duration. This 645 km altitude is referenced for compliance with Requirement 4.6-1(a) as a worst-case deorbit time scenario for a failure at this maximum altitude. Constraining this maneuver to a maximum altitude of 645 km circular results in the transfer orbit depicted in Figure 1.1. This diagram illustrates that initially the orbit altitude would be raised to the maximum of 645km, then the inclination would be adjusted from the launch insertion inclination to a value of $\sim 97.25^\circ$. This orbit will be maintained until the RAAN/LT DN has drifted to the desired value at which point the inclination will be increased to match the customer spacecraft inclination of 97.47° . Finally, the orbit altitude will be lowered to 518 km circular to match that of the customer spacecraft. The maximum altitude for this transfer orbit is above the altitude for the insertion orbit in order to use the relative precession rates between the transfer orbit and the operational orbit to assist with the RAAN change (*i.e.*, at the maximum altitude in the transfer orbit, where it is above the insertion orbit, the relative rate of precession for the transfer orbit compared to the operational orbit is more negative, assisting with the required positive RAAN change).

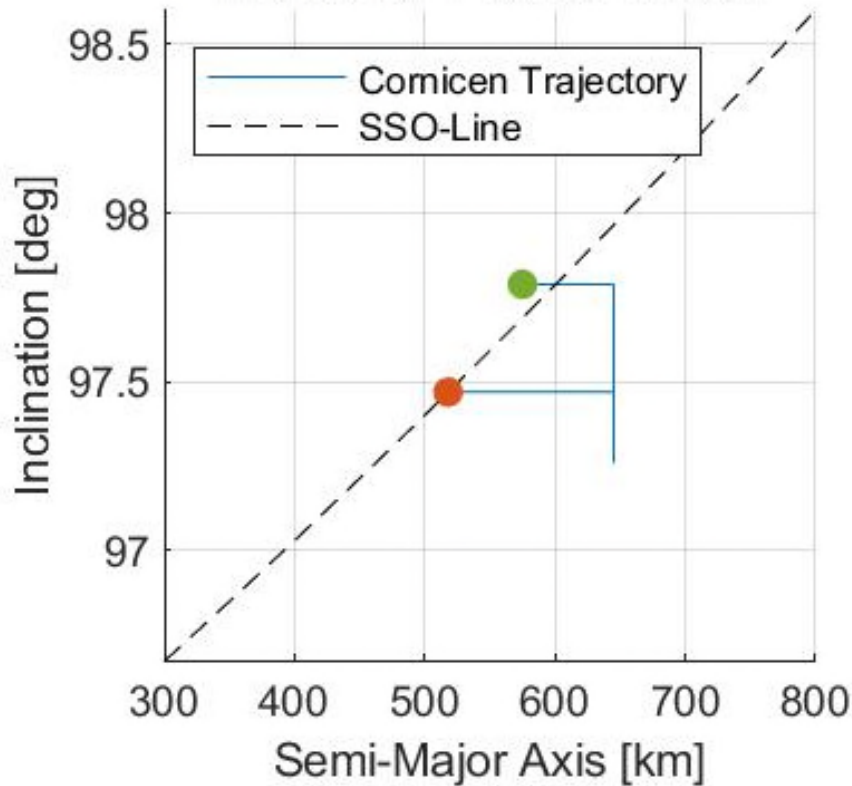


Figure 1.1. Maximum Altitude Constrained Approach Maneuver

The approach maneuver can also require an altitude drop to produce a positive RAAN change. The maximum positive RAAN change exists for an insertion orbit LTDN of 10:00 and an operational orbit LTDN of 10:45, giving an LTDN change of +45 minutes, corresponding to a RAAN change of +11.25 degrees. The worst-case altitude for the insertion orbit is 575 km SSO with a corresponding worst-case inclination of 97.69 degrees. Given these conditions, an approach that optimizes for minimum transfer time between the insertion orbit and the operational orbit can be calculated.

This transfer orbit is depicted in Figure 1.2. The minimum altitude for this transfer orbit is below the altitude for the operational orbit in order to use the relative precession rates between the transfer orbit and the operational orbit to assist with the RAAN change (i.e., at the minimum altitude in the transfer orbit, where it is below the operational orbit, the relative rate of precession for the transfer orbit compared to the operational orbit is positive, assisting with the required positive RAAN change). The higher inclination is maintained for a large portion of the transfer orbit as this greater inclination also contributes to a greater positive relative precession rate for the transfer orbit. To avoid the ISS, Cornicen's minimum altitude will not be reduced below 460 km.

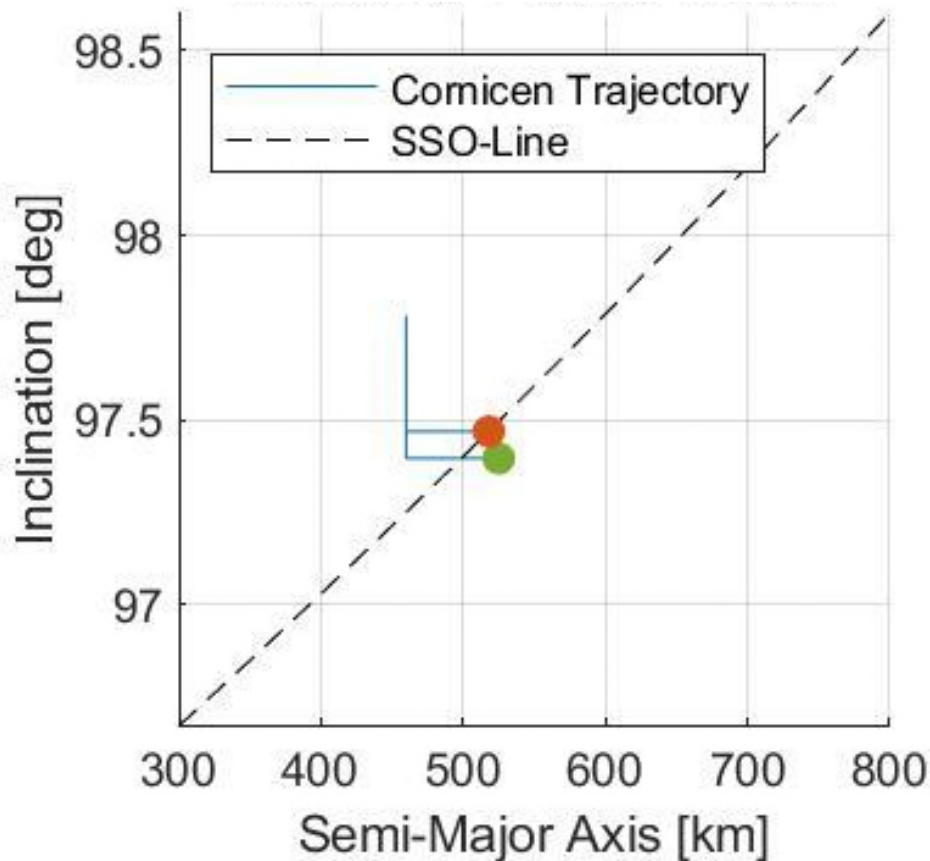


Figure 1.2. Minimum Altitude Constrained Approach Maneuver

Spacecraft maneuver capability, including attitude and orbit control and time period during which capabilities will be exercised:

The spacecraft is equipped with a three-axis active control system and a field-emission electric propulsion system. Attitude and orbit control will be exercised during the entire mission duration, including the approach maneuver from insertion to operational orbit and maintenance at the operational orbit. The attitude and orbit control will be exercised for the orbital transfer and operational orbit station-keeping and collision avoidance.

Reason for selection of operational orbit(s) (such as ground track, SSO, GEO sync, instrument resolution, or co-locate with other spacecraft):

The operational orbit is a 518 km SSO, with SSO inclination of 97.47 degrees, at LTDN 10:30 \pm 15 minutes. This precise operational orbit is necessary for co-location with a customer spacecraft, as discussed in more detail below.

Identification of any interaction or potential physical interference with other operational spacecraft: Cornicen will at all times maintain a minimum physical distance of approximately 50 to 100 km away from the partner satellite to ensure operational safety.

ODAR SECTION 2: SPACECRAFT DESCRIPTION

Clear overall spacecraft dimensions: The physical dimensions of the Cornicen spacecraft bus are 39 cm x 39 cm x 60 cm.

Total spacecraft mass at launch, including all propellants and fluids: ~39.8 kg

Dry mass of spacecraft at launch, minus consumables and propellants: ~38.5 kg

Identification, including type, mass and pressure, of all fluids (liquids and gases) planned to be on board, excluding fluids in sealed heat pipes. Description of all fluid systems, including size, type, and qualifications of fluid containers such as propellant and pressurization tanks, including pressurized batteries:

No fluid in pressurized batteries.

The Indium ingot in the Enpulsion system is launched as a solid and only melted once on-orbit. The unit is delivered from the manufacturer already loaded with propellant.

Description of all propulsion systems (e.g., cold gas, mono-propellant, bi-propellant, solid propellant, electric, nuclear):

Propulsion system includes one Enpulsion Micro R³ 100. The Enpulsion Micro R³ 100 is a field-emission electric propulsion (FEEP) thruster with 1.3 kg of propellant mass, a total impulse greater than 50 kNs, a nominal thrust of 1.0 mN, a dynamic thrust range of 75 μ N to 1.45 mN, and a dynamic specific impulse range of 1500 s to 6000 s. The Enpulsion Micro R³ has greater than 250 hours of in-space operations. The propellant for this system is non-toxic indium, which is inert, and is stored as a solid, unpressurized, and without chemical energy. The indium propellant is solid at room temperature and remains in solid state when contained in the unpowered Enpulsion Micro thruster system. The FEEP technology does not release any persistent liquids.

The indium in the Enpulsion system is a solid ingot of metal. The only way for the indium to flow out of the thruster is for the heater to melt the ingot and allow it to flow out of the thrust head. The path from the tank to space requires the indium to pass through extremely small pores in the tungsten emitter tips, and the only way for the indium to flow through these tips is for it to be vaporized into extremely small particles (i.e., only a few atoms wide).

Description of all active and/or passive attitude control systems with an indication of the normal attitude of the spacecraft with respect to the velocity vector:

Attitude determination and control system includes three reaction wheels, a reaction wheel controller, three magnetorquer rods, torque rod drivers, an ADCS controller, two star trackers, three 3-axis magnetometers, two GPS antennas, a GPS receiver, a 3-axis inertial measurement unit and six coarse sun sensors. Nominally, when not pointing an antenna or payload, the spacecraft will fly with its solar panels in a sun-pointing configuration.

Description of any range safety or other pyrotechnic devices: None.

Space vehicle separation will be accomplished using the MKII Lightband debris-free low-shock release system.

Solar array deployment will also be achieved using mechanisms that do not release any debris.

Description of the electrical generation and storage system:

The spacecraft uses two sets of 7x series Lithium Ion Samsung 48G 21700 4800mAh 4.8A batteries. Battery assembly will be charged at time of integration and two deployable solar arrays and one fixed solar array will recharge the assembly on-orbit. Power management and distribution electronics onboard the spacecraft control the charge of the battery and flow of power to other spacecraft elements.

Identification of any other sources of stored energy not noted above: None

Identification of any radioactive material on board: None

Address the trackability of the spacecraft. Spacecraft operating in low-Earth orbit will be presumed trackable if each individual spacecraft is 10 cm or larger in its smallest dimension, excluding deployable components: The spacecraft is in low-Earth orbit and has a smallest dimension measuring more than 10 cm. Thus, it is trackable.

The statement shall also disclose the following:

How the operator plans to identify the spacecraft following deployment and whether spacecraft tracking will be active or passive:

Spacecraft tracking will be both active and passive. Prior to deployment an initial ephemeris is produced by the launch service that will be used to schedule a contact time

window with the ground station. The accuracy of the ephemeris will depend on the launch vehicle's performance, delays, and deployment times. A more accurate ephemeris will be made available shortly after deployment. After deployment when the satellite asserts complete control using the ADCS, the Cornicen satellite engages its GPS unit, collects positional states, and establishes contact with the nearest available Hedron-contracted ground station. The collected data will be used to identify the spacecraft from TLE once they become available from the 18th Space Control Squadron.

Whether, prior to deployment, the spacecraft will be registered with the 18th Space Control Squadron or successor entity: Prior to deployment the spacecraft will be registered with the 18th Space Control Squadron via their Satellite Registration Form and Space Situational Awareness (SSA) Sharing Agreement.

The extent to which the spacecraft operator plans to share information regarding initial deployment, ephemeris, and/or planned maneuvers with the 18th Space Control Squadron or successor entity, other entities that engage in space situational awareness or space traffic management functions, and/or other operators:

Hedron intends to provide the 18th Space Control Squadron with information regarding initial deployment, owner/operator (O/O) ephemeris, and planned maneuvers. This is intended to provide better space situational awareness and improve the accuracy of conjunction analysis (CA). The higher accuracy O/O ephemeris will improve the quality of the CA and reduce the frequency of Conjunction Data Messages (CDMs). Hedron intends to share ephemeris with other operators on an as-needed basis.

With respect to the partner spacecraft operator, during formation flying, Hedron will engage in periodic discussions and have established operating protocols to ensure flight safety for both spacecraft.

Description of any planned proximity operations or docking with other spacecraft in LEO or GEO, including the controls that will be used to mitigate the risk of a collision that could generate debris or prevent planned later passivation or disposal activities for either spacecraft:

Cornicen will perform proximity operations in the form of formation-flying with a customer spacecraft. This will occur in the operational 518 km SSO. A control box method is used to constrain Cornicen to a safe range of approximately 50 km to 100 km distance from the customer satellite when in formation. The relative orbit within the control box is designed to provide safety with the free-flight trajectories separating the two vehicles instead of creating a fly-by scenario. This control box around the 518 km SSO defines the expected variance of and accuracy with which in-plane orbital parameters (*e.g.*, apogee and perigee) will be maintained. The out-of-plane orbital

parameters (e.g., inclination and RAAN) will also be maintained within a given tolerance.

Cornicen uses its three-axis active control system to control its drag profile, in turn controlling Cornicen's drag relative to that of the customer vehicle. Cornicen uses its electric propulsion system to reposition from the ending edge of the control box to the beginning of it, maintaining the control box. The relative position of the two spacecraft is monitored by both spacecraft operators, the mission operations centers for the two spacecraft are connected for communications, and both are equipped with collision avoidance and propulsion capabilities to ensure no collision occurs. Figure 2.1 shows the case of Cornicen formation flying with a higher relative drag compared to the partner spacecraft.

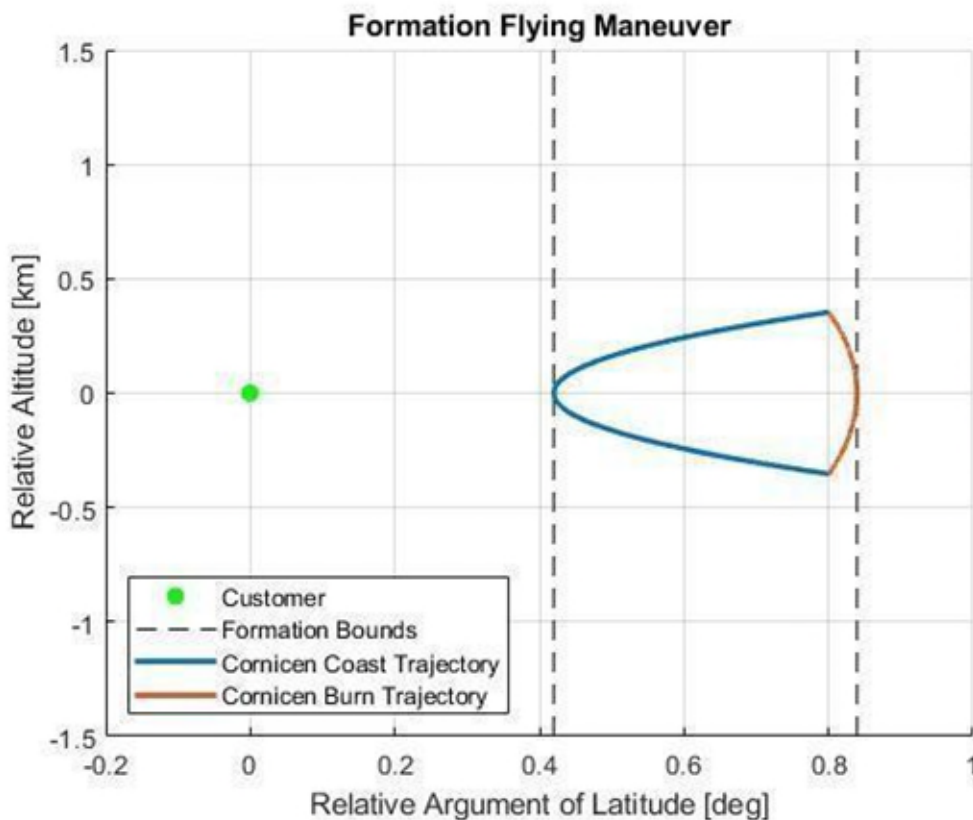


Figure 2.1. Cornicen Formation-Flying with Higher Relative Drag

ODAR SECTION 3: ASSESSMENT OF SPACECRAFT DEBRIS RELEASED DURING NOMINAL OPERATIONS

Identification of any object > 1mm expected to be released from spacecraft after launch: None

Rationale/necessity for release of each object: N/A

Time of release of each object, relative to launch time: N/A

Release velocity of each object with respect to spacecraft: N/A

Expected orbital parameters of each object after release: N/A

Calculated orbital lifetime of each object, including time spent in Low Earth Orbit:
N/A

Assessment of spacecraft compliance with ODAR Requirements 4.3-1 and 4.3-2 (per DAS v3.1.2):

- 4.3-1, Mission Related Debris Passing Through LEO: COMPLIANT
- 4.3-2, Mission Related Debris Passing Through GEO: COMPLIANT

ODAR SECTION 4: ASSESSMENT OF SPACECRAFT INTENTIONAL BREAKUPS AND POTENTIAL FOR EXPLOSION

Potential causes of spacecraft breakup during deployment and mission operations:

There are no credible causes of spacecraft breakup during nominal deployment and mission operations.

Summary of failure modes and effects analyses of all credible failure modes that may lead to an accidental explosion:

The spacecraft has no chemical propellants or pressurized vessels. The battery safety systems are discussed in the assessment of spacecraft compliance with ODAR requirement 4.4-1 and describe combined faults required for the mutually exclusive failures that lead to battery venting. The batteries are space qualified and equipped with a safety vent feature that vents excessive pressure build-up, precluding explosions.

Detailed plan for any designed spacecraft breakup: None

List of components which are passivated at End of Mission (EOM). List includes method of passivation and amount which cannot be passivated:

The following components will be passivated at EOM:

- Solar panels - method of passivation: disconnected from the electrical power system (EPS) at EOM using an electronic switch.
- Batteries - method of passivation: will be passivated at EOM. Amount that cannot be passivated: the energy left in the batteries after the EPS cut-off is calculated to be approximately 14.4 watt-hours.

- Reaction wheels - method of passivation: will be switched off and power removed at EOM. Amount that cannot be passivated: all kinetic energy will be dissipated before reentry.

Items which are required to be passivated, but cannot be due to their design: None

Assessment of spacecraft compliance with ODAR requirements 4.4-1 - 4.4-4:

4.4-1: Limited the risk to other space systems from accidental explosions during deployment and mission operations while in orbit around Earth or the Moon:

- Required Probability: 0.001 – COMPLIANT; expected probability of 0.000

Battery explosion:

- Effect: All failure modes below might result in battery explosion with the possibility of orbital debris generation. However, in the unlikely event that a battery cell does explosively rupture, the small size, mass, and potential energy, of these small batteries is such that while the spacecraft could be expected to vent gases, most debris from the battery rupture should be contained within the spacecraft due to the lack of penetration energy due to the multiple enclosures surrounding the batteries.
- Probability: Extremely Low. It is believed to be less than 0.01% given that multiple independent (not common mode) faults must occur for each failure mode to cause the ultimate effect (explosion). Each battery cell is UL/UN certified with individual over-voltage and over-current protection. Identical batteries have been flown on all spacecraft from the bus manufacturer (Astro Digital). Even in extreme cases (such as a launch vehicle hydrazine explosion in proximity to the spacecraft), the batteries showed no signs of damage or degradation.

Supporting Rationale:

- Failure Mode 1: Internal cell short circuit
 - Mitigation 1: Protoflight level sine burst, sine and random vibration in three axes of the spacecraft, thermal vacuum cycling of the spacecraft and extensive functional testing followed by maximum system rate-limited charge and discharge cycles were performed to prove that no internal short circuit sensitivity exists. Additional environmental and functional testing of the batteries at the power subsystem vendor facilities were also conducted on the batteries at the component level.
 - Combined faults required for realized failure: Environmental testing AND functional charge/discharge tests must both be ineffective in discovery of the failure mode.
- Failure Mode 2: Internal thermal rise due to high load discharge rate.

- Mitigation 2: Battery cells were tested in lab for high load discharge rates in a variety of flight-like configurations to determine the feasibility of an out-of-control thermal rise in the cell. Cells were also tested in a hot, thermal vacuum environment (5 cycles at 50C, then to -20C) in order to test the upper limit of the cells' capability. No failures were observed or identified via satellite telemetry or via external monitoring circuitry.
- Combined faults required for realized failure: Spacecraft thermal design must be incorrect AND external over-current detection and disconnect function must fail to enable this failure mode.
- Failure Mode 3: Excessive discharge rate or short-circuit due to external device failure or terminal contact with conductors not at battery voltage levels (due to abrasion or inadequate proximity separation).
 - Mitigation 3: This failure mode is negated by:
 - a) Qualification tested short circuit protection on each external circuit,
 - b) Design of battery packs and insulators such that no contact with nearby board traces is possible without being caused by some other mechanical failure,
 - c) Observation of such other mechanical failures by protoflight level environmental tests (sine burst, random vibration, thermal cycling, and thermal-vacuum test).
 - Combined faults required for realized failure: An external load must fail/short-circuit AND external over-current detection and disconnect function must all occur to enable this failure mode.
- Failure Mode 4: Inoperable vents
 - Mitigation 4: Battery venting is not inhibited by the battery holder design or the spacecraft design. The battery can vent gases to the external environment.
 - Combined faults required for realized failure: The cell manufacturer OR the satellite integrator fails to install proper venting.
- Failure Mode 5: Crushing
 - Mitigation 5: Failure mode prevented by design. No moving parts near the battery assembly.
 - Combined faults required for realized failure: A catastrophic failure must occur in an external system AND the failure must cause a collision sufficient to crush the batteries leading to an internal short circuit AND the satellite must be in a naturally sustained orbit at the time the crushing occurs.
- Failure Mode 6: Low level current leakage or short-circuit through battery pack case or due to moisture-based degradation of insulators
 - Mitigation 6: These modes are negated by:

- a) Battery holder/case design made of non-conductive plastic, and
- b) Operation in vacuum such that no moisture can affect insulators.
- Combined faults required for realized failure: Abrasion or piercing of circuit board coating or battery wire harness insulator AND dislocation of battery packs AND failure of battery terminal insulators AND failure to detect such failures in environmental tests must occur to result in this failure mode.
- Failure Mode 7: Excess battery cell temperature due to orbital environment and high discharge combined.
 - Mitigation 7: The spacecraft thermal design will negate this possibility. Thermal rise has been analyzed in combination with space environment temperatures showing that batteries do not exceed normal allowable operating temperatures under a variety of modeled cases, including worst case orbital scenarios. Analysis shows these temperatures to be well below temperatures of concern for explosions.
 - Combined faults required for realized failure: incorrect thermal analysis AND thermal design AND mission simulations in thermal-vacuum chamber testing AND over-current monitoring and control must all fail for this failure mode to occur.

4.4-2: Design for passivation after completion of mission operations while in orbit about Earth or the Moon:

- Compliance Statement: Circuits are designed to allow for the disconnection of all solar panel power. In this EOM mode, the battery discharges to the low voltage cutoff within a few days. At the low voltage cutoff, the total battery energy remaining is approximately 14.4 watt-hours. Once in EOM mode with low battery voltage cutoff the satellite cannot be recovered.

4.4-3: Limiting the long-term risk to other space systems from planned breakups:

- Compliance Statement: Requirement not applicable, no planned breakups

4.4-4: Limiting the short-term risk to other space systems from planned breakups:

- Compliance Statement: Requirement not applicable, no planned breakups

ODAR SECTION 5: ASSESSMENT OF SPACECRAFT POTENTIAL FOR ON-ORBIT COLLISIONS

Calculation of spacecraft probability of collision with space objects larger than 10 cm in diameter during orbital lifetime of spacecraft:

Per DAS v3.1.2, the probability of collision with large space objects is 6.5776E-06.

Assessment of spacecraft compliance with ODAR requirements 4.5-1 and 4.5-2:

- 4.5-1: Limiting debris generated by collisions with large objects when operating in Earth Orbit (per DAS v3.1.2):
 - COMPLIANT; Collision Probability < 0.001
- 4.5-2: Limiting debris generated by collisions with small objects when operating in Earth or Lunar Orbit:
 - COMPLIANT; Not applicable as the planned disposal method is via atmospheric reentry that does not require a specific orientation or drag state.

Detailed description and assessment of efficacy of any planned debris avoidance capability intended to help in meeting requirement 4.5-1:

Cornicen is equipped with a three-axis active control system and an electric propulsion system. These attitude and orbit control systems are used in tandem with situational awareness, conjunction assessment, and collision avoidance (COLA) ground operations software to maneuver the vehicle out of any potential collisions, thus reducing the collision risk to effectively zero. Cornicen will provision the required amount of propellant to enable collision avoidance given the estimated number of potential conjunction events in its operational orbit.

If at any time during the spacecraft's mission or de-orbit phase the spacecraft will operate in or transit through the orbits used by any planned or inhabitable spacecraft, including the International Space Station, describe the design and operational strategies, such as coordination, that will be used to minimize the risk of collision and avoid posing any operational constraints to the spacecraft:

There are several cases that may apply including the approach maneuver, nominal operations at operational orbit, and de-orbit phase. During the approach phase, where the spacecraft maneuvers from the insertion orbit to the operational orbit, it is possible that the spacecraft will transit through orbits used by planned or inhabited spacecraft. It is possible that the spacecraft will transit through orbits used by planned or inhabited spacecraft during nominal operations. Nominal operations for the spacecraft use formation flying with a customer vehicle, thus the operational orbit is co-inhabited by the customer spacecraft. The de-orbit phase will transfer the spacecraft from an altitude greater than that of the ISS to an altitude below that of the ISS, thus this phase may transit through an orbit used by the ISS or another spacecraft.

For the provided cases, the spacecraft is equipped with attitude control and propulsion systems and the operations center for the spacecraft has collision avoidance

operationally active, thus the collision risk during these operations is effectively zero. Additionally, when formation flying with the customer spacecraft, both spacecraft monitor their relative positions, have communicating operations centers, and have maneuvering and collision avoidance capabilities to ensure no collisions occur, as previously described. Although excess propellant is used during the de-orbit phase to assist with lowering altitude, propellant margin is reserved for potential collision avoidance maneuvers and the spacecraft is collision avoidance capable until the demise altitude. The de-orbit maneuver will use circular orbit reduction, maintaining a circular orbit to reduce the period for potential conjunction events with the ISS.

Certify that upon receipt of a space situational awareness conjunction warning, the operator will review and take all possible steps to assess the collision risk, and will mitigate the collision risk if necessary:

Upon receipt of a space situational awareness conjunction warning, the operator will attempt to contact the operator of any active spacecraft involved in such a warning, share ephemeris data and other appropriate operational information with any such operator, and modify the spacecraft's attitude, orbit, and/or operations in order to avoid a collision. If the operator of the other spacecraft does not have maneuvering capabilities or is not expedient in response, the operator for Cornicen will upload potential maneuvers via API to the 18th Space Control Squadron in order to avoid a collision. High risk events will be mitigated to at least one and a half orders of magnitude below the collision probability mitigated action threshold of $1E-5$.

ODAR SECTION 6: ASSESSMENT OF SPACECRAFT POST-MISSION DISPOSAL PLANS AND PROCEDURES

Description of spacecraft disposal option selected:

Per NASA-STD 8719.4, Cornicen will be disposed of via atmospheric reentry. The operational altitude lends to natural forces that will quickly lead to atmospheric reentry once the operations, including stationkeeping with maneuvers, are ceased.

Identification of all systems or components required to accomplish any post-mission disposal maneuvers. Plan for any spacecraft maneuvers required to accomplish post-mission disposal:

No systems, components, special maneuvers, or operations are required for post-mission disposal. The spacecraft will deorbit naturally.

However, at the end of the operational lifetime of the satellite, any remaining excess propellant will be employed to lower apogee and perigee (maintaining a circular orbit to reduce the period for potential conjunction events with the ISS) until either all

remaining propellant is consumed or the control authority of the attitude control system is reached.

Calculation of area-to-mass ratio after post-mission disposal:

The final area-to-mass ratio is calculated using the dry mass of the vehicle and the average cross-sectional area. Average cross-sectional area is calculated using the equation for estimated average cross-sectional area for non-convex shapes from NASA-STD-8179.14C: $A_{avg} = (A_{max} + A_1 + A_2)/2$, where A_{max} is the maximum cross-sectional area and A_1 and A_2 are the cross-sectional areas for the two viewing directions orthogonal to the maximum cross sectional area viewing direction.

Final area-to-mass ratio calculation:

- Spacecraft mass: 38.5 kg (dry mass)
- Cross-sectional area: 0.483 m² (average)
- Area-to-mass ratio: 0.0125 m²/kg (final)

Calculation of area-to-mass ratio for worst-case deorbit scenarios:

Additional analysis is provided for the two worst-case deorbit scenarios in the following section on ODAR requirement 4.6-1, those for deorbit time with the vehicle at the maximum altitude launch deployment with undeployed solar panels and for the vehicle at maximum altitude during the approach maneuver with deployed solar panels. The average cross-sectional area for these two cases is calculated using the same equation from NASA-STD-8179.14C as previously provided.

Undeployed area-to-mass ratio calculation:

- Spacecraft mass: 39.8 kg (wet mass)
- Cross-sectional area: 0.407 m² (average)
- Area-to-mass ratio: 0.0102 m²/kg (final)

Max approach altitude area-to-mass ratio calculation:

- Spacecraft mass: 39.8 kg (wet mass with some propellant used)
- Cross-sectional area: 0.483 m² (average)
- Area-to-mass ratio: 0.0121 m²/kg (final)

Assessment of spacecraft compliance with ODAR requirements 4.6-1 to 4.6-4:

- 4.6-1(a): Disposal for space structures in or passing through LEO:
 - COMPLIANT - The DAS prediction for orbit lifetime following a nominal three-year mission lifetime is approximately three years as shown in Figure 6.1. Accordingly, the spacecraft complies with the 25-year re-entry requirement.

- Two alternative worst-case deorbit times are considered for if the spacecraft were deployed at the worst-case altitude of 575 km and failed prior to solar panel deployment and if the spacecraft were to have a failure at the maximum altitude of 645 km along the approach trajectory with the solar panels deployed. For the former of these cases, the satellite would re-enter the Earth's atmosphere in approximately 13.1 years, while for the latter, it would re-enter in approximately 24.6 years. These are shown in Figures 6.2 and 6.3, respectively. For these worst-case scenarios, the spacecraft still complies with the 25-year re-entry requirement.

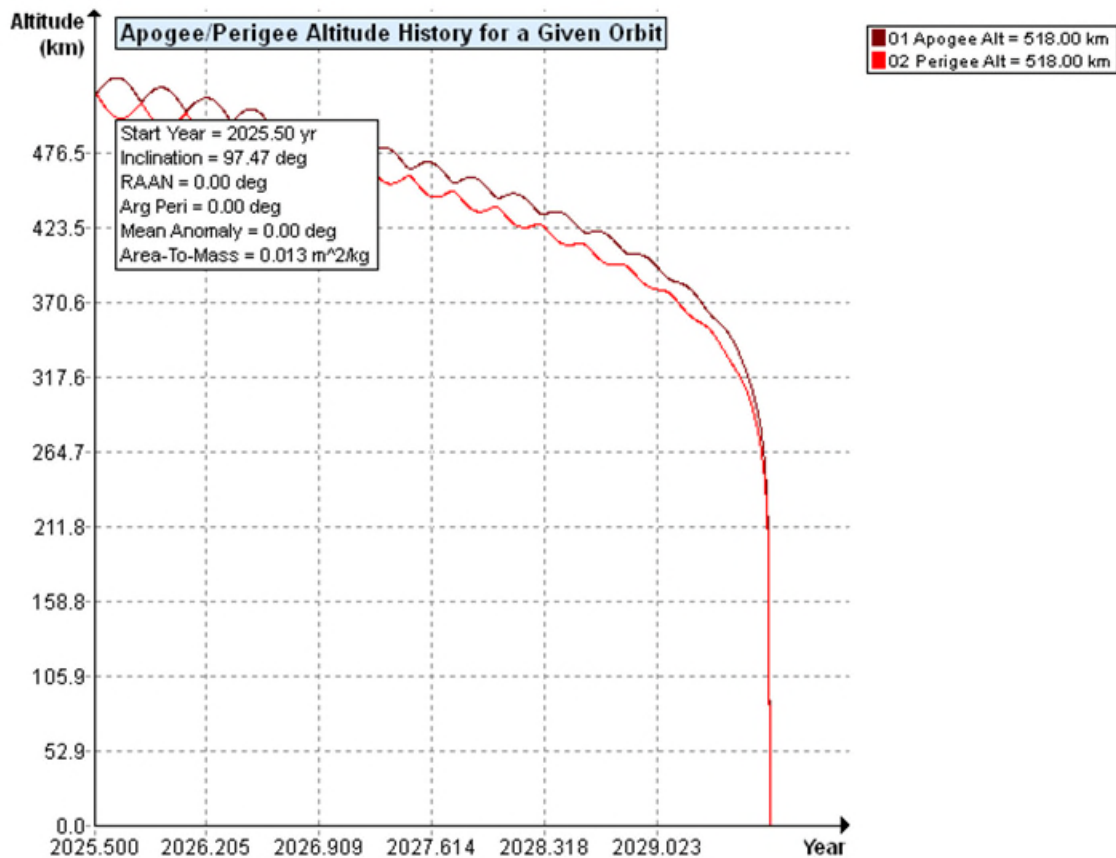


Figure 6.1. Cornicen De-orbit Altitude Profile from Operational Altitude

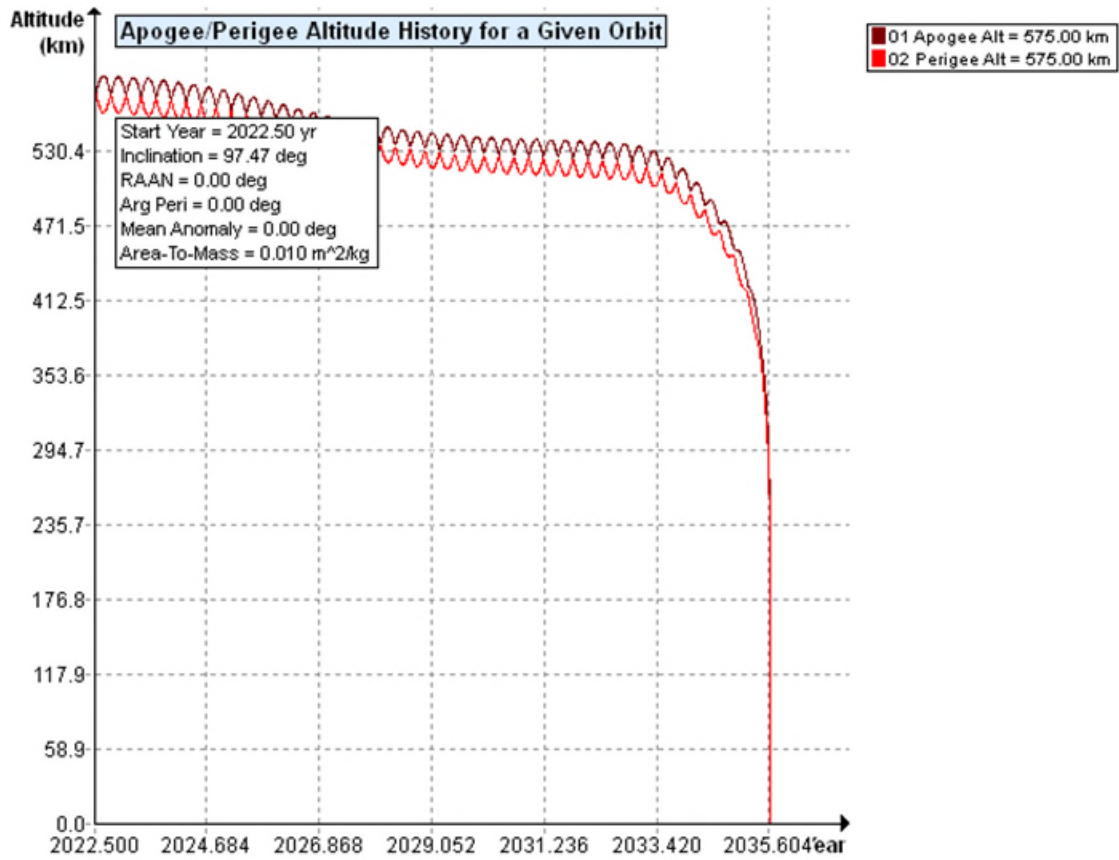


Figure 6.2. Cornicen De-orbit Altitude Profile from Worst-case Deployment Altitude – Assuming Solar Arrays Never Deploy

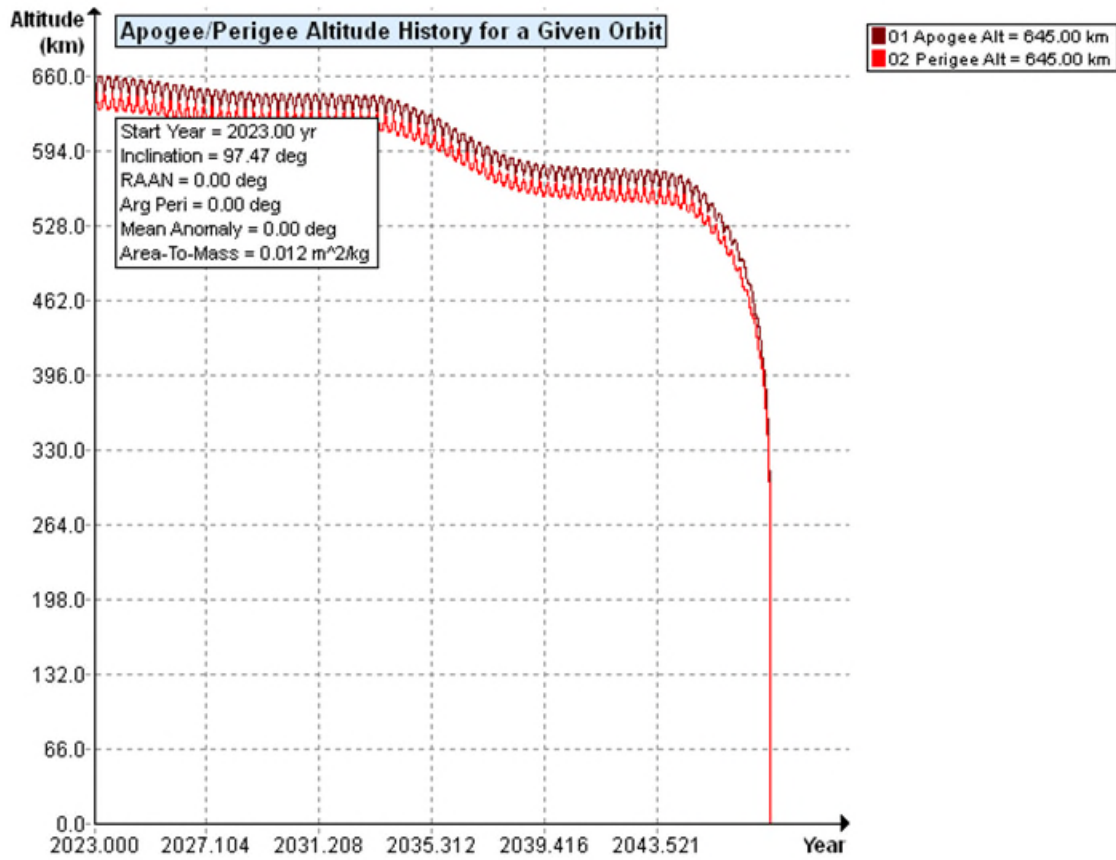


Figure 6.3. Cornicen De-orbit Altitude Profile from Worst-Case Approach Altitude

- 4.6-2: Disposal for space structure near GEO
 - Not applicable
- 4.6-3: Disposal for space structures between LEO and GEO
 - Not applicable
- 4.6-4: Reliability of post-mission disposal operations
 - The spacecraft will reenter passively without post-mission disposal operations within the required timeframe

ODAR SECTION 7: ASSESSMENT OF SPACECRAFT REENTRY HAZARDS

Summary of objects expected to survive uncontrolled reentry:

No spacecraft components are expected to survive uncontrolled reentry per the DAS v3.1.2 Casualty Risk from Reentry Debris requirement assessment. This corresponds to a 1:100,000,000 risk of human casualty and a total debris casualty area of 0.00 m².

Assessment of spacecraft compliance with ODAR requirement 4.7-1:

- 4.7-1(a): Limit the risk of human casualty from surviving debris for an uncontrolled reentry to no greater than 1 in 10,000
 - COMPLIANT per DAS v3.1.2; risk is 1 in 100,000,000
- 4.7-1(b): Not applicable, only for controlled reentry
- 4.7-1(c): Not applicable, only for controlled reentry

ODAR SECTION 8: ASSESSMENT FOR SPECIAL CLASSES OF SPACE MISSIONS

None of the special classes in this Section are applicable, except as discussed above with respect to formation flying.

DAS v3.1.2 OUTPUT FILE

12 03 2021; 06:39:56AM Processing Requirement 4.3-1: Return Status : Not Run

=====
No Project Data Available
=====

===== End of Requirement 4.3-1 =====

12 03 2021; 06:39:59AM Processing Requirement 4.3-2: Return Status : Passed

=====
No Project Data Available
=====

===== End of Requirement 4.3-2 =====

12 03 2021; 07:31:00AM Processing Requirement 4.5-1: Return Status : Passed

=====
Run Data
=====

INPUT

Space Structure Name = Greylock
Space Structure Type = Payload
Perigee Altitude = 518.000 (km)
Apogee Altitude = 518.000 (km)
Inclination = 97.471 (deg)
RAAN = 0.000 (deg)
Argument of Perigee = 0.000 (deg)
Mean Anomaly = 0.000 (deg)
Final Area-To-Mass Ratio = 0.0125 (m²/kg)
Start Year = 2022.500 (yr)
Initial Mass = 39.800 (kg)
Final Mass = 38.500 (kg)
Duration = 3.000 (yr)
Station-Kept = True
Abandoned = True

OUTPUT

Collision Probability = 6.5776E-06

Returned Message: Normal Processing
Date Range Message: Normal Date Range
Status = Pass

=====

===== End of Requirement 4.5-1 =====

12 03 2021; 07:31:02AM Processing Requirement 4.6 Return Status : Passed

=====

Project Data

=====

****INPUT****

Space Structure Name = Greylock
Space Structure Type = Payload

Perigee Altitude = 518.000000 (km)
Apogee Altitude = 518.000000 (km)
Inclination = 97.471000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.012500 (m²/kg)
Start Year = 2022.500000 (yr)
Initial Mass = 39.800000 (kg)
Final Mass = 38.500000 (kg)
Duration = 3.000000 (yr)
Station Kept = True
Abandoned = True
PMD Perigee Altitude = 518.000000 (km)
PMD Apogee Altitude = 518.000000 (km)
PMD Inclination = 97.471000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)

****OUTPUT****

Suggested Perigee Altitude = 518.000000 (km)
Suggested Apogee Altitude = 518.000000 (km)
Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2029 (yr)
Requirement = 61
Compliance Status = Pass

=====

===== End of Requirement 4.6 =====

12 03 2021; 07:31:12AM *****Processing Requirement 4.7-1
Return Status : Passed

*****INPUT*****

Item Number = 1

name = Greylock
quantity = 1
parent = 0
materialID = 9
type = Box
Aero Mass = 38.500000
Thermal Mass = 38.500000
Diameter/Width = 0.370000
Length = 0.590000
Height = 0.370000

name = Data Power Module
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 2.300000
Thermal Mass = 2.300000
Diameter/Width = 0.100000
Length = 0.189000
Height = 0.090000

name = Direct Energy Pack
quantity = 2
parent = 1
materialID = 8
type = Box
Aero Mass = 2.200000
Thermal Mass = 2.200000
Diameter/Width = 0.170000

Length = 0.194000
Height = 0.124000

name = Reaction wheel
quantity = 3
parent = 1
materialID = 8
type = Box
Aero Mass = 0.250000
Thermal Mass = 0.250000
Diameter/Width = 0.065000
Length = 0.077000
Height = 0.038000

name = Torque rod
quantity = 3
parent = 1
materialID = 46
type = Cylinder
Aero Mass = 0.490000
Thermal Mass = 0.490000
Diameter/Width = 0.020000
Length = 0.300000

name = NX payload computer
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.300000
Thermal Mass = 0.300000
Diameter/Width = 0.085000
Length = 0.105000
Height = 0.030000

name = Star tracker
quantity = 1
parent = 1
materialID = 8
type = Cylinder
Aero Mass = 0.750000
Thermal Mass = 0.750000
Diameter/Width = 0.075000
Length = 0.120000

name = Gyro
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.080000
Thermal Mass = 0.080000
Diameter/Width = 0.038000
Length = 0.045000
Height = 0.022000

name = Torque board
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.030000
Thermal Mass = 0.030000
Diameter/Width = 0.030000
Length = 0.090000
Height = 0.010000

name = GPS antenna
quantity = 2
parent = 1
materialID = 8
type = Box
Aero Mass = 0.030000
Thermal Mass = 0.030000
Diameter/Width = 0.030000
Length = 0.030000
Height = 0.020000

name = UHF antenna
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.030000
Thermal Mass = 0.030000
Diameter/Width = 0.030000
Length = 0.050000
Height = 0.010000

name = S-band RX antenna
quantity = 3
parent = 1
materialID = 8
type = Box
Aero Mass = 0.030000
Thermal Mass = 0.030000
Diameter/Width = 0.040000
Length = 0.060000
Height = 0.010000

name = Y-
quantity = 1
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 1.520000
Thermal Mass = 1.520000
Diameter/Width = 0.350000
Length = 0.560000

name = X+
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 2.450000
Thermal Mass = 2.450000
Diameter/Width = 0.350000
Length = 0.350000
Height = 0.300000

name = X-
quantity = 1
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 1.250000
Thermal Mass = 1.250000
Diameter/Width = 0.350000
Length = 0.350000

name = Z+

quantity = 1
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 1.520000
Thermal Mass = 1.520000
Diameter/Width = 0.350000
Length = 0.560000

name = Y+
quantity = 1
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 1.520000
Thermal Mass = 1.520000
Diameter/Width = 0.350000
Length = 0.560000

name = Z-
quantity = 1
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 1.520000
Thermal Mass = 1.520000
Diameter/Width = 0.350000
Length = 0.560000

name = Reaction wheel enclosure
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.300000
Thermal Mass = 0.300000
Diameter/Width = 0.350000
Length = 0.560000
Height = 0.010000

name = MLB
quantity = 1
parent = 1
materialID = 9

type = Cylinder
Aero Mass = 0.580000
Thermal Mass = 0.580000
Diameter/Width = 0.240000
Length = 0.300000

name = Main Solar Panel
quantity = 3
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 1.990000
Thermal Mass = 1.990000
Diameter/Width = 0.320000
Length = 0.450000

name = Propulsion Thruster
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 1.970000
Thermal Mass = 1.290000
Diameter/Width = 0.120000
Length = 0.140000
Height = 0.092600

name = Propulsion Reservoirs
quantity = 4
parent = 22
materialID = 54
type = Cylinder
Aero Mass = 0.170000
Thermal Mass = 0.170000
Diameter/Width = 0.042500
Length = 0.068400

name = Propulsion PPU
quantity = 1
parent = 1
materialID = -2
type = Box
Aero Mass = 0.630000
Thermal Mass = 0.630000

Diameter/Width = 0.120000
Length = 0.140000
Height = 0.034900

name = Primary Payload 1: GEOLink Antenna
quantity = 1
parent = 1
materialID = -1
type = Box
Aero Mass = 0.140000
Thermal Mass = 0.140000
Diameter/Width = 0.082000
Length = 0.222000
Height = 0.009600

name = Primary Payload 1: GEOLink Transceiver
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 1.100000
Thermal Mass = 1.100000
Diameter/Width = 0.096000
Length = 0.125000
Height = 0.070000

name = Primary Payload 2: SDX
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 2.500000
Thermal Mass = 2.500000
Diameter/Width = 0.197900
Length = 0.257800
Height = 0.038860

name = Primary Payload 2: uTX
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.550000
Thermal Mass = 0.550000

Diameter/Width = 0.092700
Length = 0.106700
Height = 0.027690

name = Primary Payload 2: uRX
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.550000
Thermal Mass = 0.550000
Diameter/Width = 0.092700
Length = 0.106700
Height = 0.027690

name = Primary Payload 2: Primary S-band Antenna
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.580000
Thermal Mass = 0.580000
Diameter/Width = 0.190000
Length = 0.250000
Height = 0.006430

name = Primary Payload 2: Primary X-band Antenna
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.210000
Thermal Mass = 0.210000
Diameter/Width = 0.112000
Length = 0.132000
Height = 0.005510

name = Primary Payload 2: Second S-band Antenna
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.190000
Thermal Mass = 0.190000

Diameter/Width = 0.091400
Length = 0.091400
Height = 0.010160

name = Primary Payload 2: Second X-band Antenna
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.210000
Thermal Mass = 0.210000
Diameter/Width = 0.112000
Length = 0.132000
Height = 0.005510

name = Primary Payload 2: Ka-band Antenna
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.360000
Thermal Mass = 0.360000
Diameter/Width = 0.074930
Length = 0.074930
Height = 0.025400

name = Secondary Payload 1: OISL
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 2.000000
Thermal Mass = 1.940000
Diameter/Width = 0.100000
Length = 0.200000
Height = 0.100000

name = Filter substrate
quantity = 2
parent = 35
materialID = -3
type = Flat Plate
Aero Mass = 0.010000
Thermal Mass = 0.010000

Diameter/Width = 0.040000
Length = 0.040000

name = Beam Splitter substrate
quantity = 1
parent = 35
materialID = -4
type = Flat Plate
Aero Mass = 0.012000
Thermal Mass = 0.012000
Diameter/Width = 0.050000
Length = 0.050000

name = Aspheric Lens
quantity = 2
parent = 35
materialID = -4
type = Flat Plate
Aero Mass = 0.007000
Thermal Mass = 0.007000
Diameter/Width = 0.030000
Length = 0.030000

name = Laser Fiber Collimator
quantity = 1
parent = 35
materialID = 66
type = Box
Aero Mass = 0.012000
Thermal Mass = 0.012000
Diameter/Width = 0.010000
Length = 0.040000
Height = 0.010000

name = Mirror
quantity = 1
parent = 35
materialID = 71
type = Flat Plate
Aero Mass = 0.002000
Thermal Mass = 0.002000
Diameter/Width = 0.020000
Length = 0.020000

name = Secondary Payload 2: C-band Antenna
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.820000
Thermal Mass = 0.820000
Diameter/Width = 0.097030
Length = 0.204000
Height = 0.062990

name = Payload Harness
quantity = 1
parent = 1
materialID = 19
type = Box
Aero Mass = 0.180000
Thermal Mass = 0.180000
Diameter/Width = 0.050000
Length = 0.050000
Height = 0.010000

name = Primary Payload 2: Power Amplifier S-band
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.120000
Thermal Mass = 0.120000
Diameter/Width = 0.044200
Length = 0.067560
Height = 0.015750

name = Primary Payload 2: Power Amplifier X-band
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.120000
Thermal Mass = 0.120000
Diameter/Width = 0.044200
Length = 0.067560
Height = 0.015750

name = Primary Payload 2: Power Amplifier Ka-band
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.120000
Thermal Mass = 0.120000
Diameter/Width = 0.044200
Length = 0.067560
Height = 0.015750

*****OUTPUT****

Item Number = 1

name = Greylock
Demise Altitude = 77.994484
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Data Power Module
Demise Altitude = 68.297615
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Direct Energy Pack
Demise Altitude = 70.176575
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Reaction wheel
Demise Altitude = 73.781296
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Torque rod
Demise Altitude = 67.157768
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = NX payload computer
Demise Altitude = 74.098396
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Star tracker
Demise Altitude = 70.615448
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Gyro
Demise Altitude = 74.685295
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Torque board
Demise Altitude = 76.951630
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = GPS antenna
Demise Altitude = 75.898727
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = UHF antenna
Demise Altitude = 76.301529
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = S-band RX antenna
Demise Altitude = 76.755127
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Y-
Demise Altitude = 75.141846

Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = X+
Demise Altitude = 74.883018
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = X-
Demise Altitude = 74.262886
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Z+
Demise Altitude = 75.141846
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Y+
Demise Altitude = 75.141846
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Z-
Demise Altitude = 75.141846
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Reaction wheel enclosure
Demise Altitude = 77.481155
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = MLB
Demise Altitude = 76.755089
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Main Solar Panel
Demise Altitude = 74.577560
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Propulsion Thruster
Demise Altitude = 71.683540
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Propulsion Reservoirs
Demise Altitude = 63.692013
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Propulsion PPU
Demise Altitude = 72.413155
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 1: GEOLink Antenna
Demise Altitude = 77.369835
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 1: GEOLink Transceiver
Demise Altitude = 70.135071
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: SDX
Demise Altitude = 67.902550
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: uTX
Demise Altitude = 71.216751
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: uRX
Demise Altitude = 71.216751
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: Primary S-band Antenna
Demise Altitude = 74.800186
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: Primary X-band Antenna
Demise Altitude = 75.315224
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: Second S-band Antenna
Demise Altitude = 74.464371
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: Second X-band Antenna
Demise Altitude = 75.315224
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: Ka-band Antenna
Demise Altitude = 71.281013
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Secondary Payload 1: OISL
Demise Altitude = 70.371101

Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Filter substrate
Demise Altitude = 0.000000
Debris Casualty Area = 0.819200
Impact Kinetic Energy = 1.020041

name = Beam Splitter substrate
Demise Altitude = 70.170479
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Aspheric Lens
Demise Altitude = 70.110077
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Laser Fiber Collimator
Demise Altitude = 0.000000
Debris Casualty Area = 0.384400
Impact Kinetic Energy = 5.790510

name = Mirror
Demise Altitude = 69.737274
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Secondary Payload 2: C-band Antenna
Demise Altitude = 73.683113
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Payload Harness
Demise Altitude = 72.779068
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: Power Amplifier S-band
Demise Altitude = 74.391251
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: Power Amplifier X-band
Demise Altitude = 74.391251
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

name = Primary Payload 2: Power Amplifier Ka-band
Demise Altitude = 74.391251
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

===== End of Requirement 4.7-1 =====

12 03 2021; 10:15:01AM Science and Engineering - Orbit Lifetime/Dwell Time

INPUT

Start Year = 2023.000000 (yr)
Perigee Altitude = 645.000000 (km)
Apogee Altitude = 645.000000 (km)
Inclination = 97.471000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Area-To-Mass Ratio = 0.012100 (m²/kg)

OUTPUT

Orbital Lifetime from Startyr = 24.613279 (yr)
Time Spent in LEO during Lifetime = 24.613279 (yr)
Last year of Propagation = 2047 (yr)
Returned Error Message: Object reentered

12 03 2021; 10:19:32AM Science and Engineering - Orbit Lifetime/Dwell Time

INPUT

Start Year = 2022.500000 (yr)
Perigee Altitude = 575.000000 (km)
Apogee Altitude = 575.000000 (km)
Inclination = 97.471000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Area-To-Mass Ratio = 0.010200 (m²/kg)

****OUTPUT****

Orbital Lifetime from Startyr = 13.092402 (yr)
Time Spent in LEO during Lifetime = 13.092402 (yr)
Last year of Propagation = 2035 (yr)
Returned Error Message: Object reentered

12 03 2021; 10:20:09AM Science and Engineering - Orbit Lifetime/Dwell Time

****INPUT****

Start Year = 2025.000000 (yr)
Perigee Altitude = 518.000000 (km)
Apogee Altitude = 518.000000 (km)
Inclination = 97.471000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Area-To-Mass Ratio = 0.012500 (m²/kg)

****OUTPUT****

Orbital Lifetime from Startyr = 3.427789 (yr)
Time Spent in LEO during Lifetime = 3.427789 (yr)
Last year of Propagation = 2028 (yr)
Returned Error Message: Object reentered